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HIGH-SPEED TESTS OF AN AIRFOIL SECTION CAMBERED TO HAVE CRITICAL MACH NUMBERS HIGHER THAN THOSE ATTAINABLE WITH A UNIFORM-LOAD MEAN LINE

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SUMMARY

High-speed wind-tunnel tests have been made to determine the aerodynamic characteristics of an NACA 6-series airfoil section especially cambered to have critical Mach numbers higher than those for an airfoil having the same design lift coefficient with a uniform-load type of mean camber line. Section coefficients of lift, drag, and pitching moment for the airfoil, designated as a modified NACA $66_{(109)}$-210 section with a modified mean camber line, are presented for angles of attack through the lift stall at Mach numbers up to approximately 0.9. Comparisons are made between the characteristics of the modified airfoil and those of the NACA 66-210 airfoil with a uniform-load type of mean camber line.

The test results indicate most of the characteristics of the modified NACA $66_{(109)}$-210 airfoil to be essentially the same as those of the NACA 66-210 (a = 1.0) airfoil. The especially cambered airfoil exhibits slightly more favorable lift- and drag-divergence characteristics, however, than the NACA 66-210 (a = 1.0) airfoil, the former having divergence Mach numbers approximately 0.01 higher than those of the latter over most of the useful lift-coefficient range. This small difference roughly corresponds to the difference in the critical speeds of the two airfoil sections.

INTRODUCTION

The principal objective in the design of airfoil sections for high-speed applications is the realization of high force-divergence Mach numbers. For design purposes, however, the airfoil critical Mach number is a more useful parameter than the force-divergence Mach number because it is directly under the control of the designer. There being a qualitative correspondence between the critical Mach number and the lift- and drag-divergence Mach numbers of an airfoil, the higher critical speed airfoils having higher lift- and drag-divergence Mach numbers, efforts to evolve suitable airfoil sections for high speeds accordingly have been concentrated on the development of airfoils having high critical Mach numbers. Design methods have progressed sufficiently in this respect that for an airfoil section of given thickness-chord ratio the critical Mach number for any specified lift coefficient may be brought very close to the maximum theoretically possible. In the design of airfoils for various practical applications, of course, some compromises are necessary which generally involve sacrifices in critical speed below the maximum attainable. In particular, the design must be such as to permit a rate of pressure recovery near the trailing edge which will result in a practicable airfoil. Further, in general applications, high critical speeds are desired over a range of lift coefficients. This objective is accomplished at some expense to the highest critical Mach number by providing at the design lift coefficient a fairly steep gradient of falling pressures from the leading edge to the minimum pressure position with a consequently decreased minimum pressure. The steeper pressure gradient permits, within limits, a variation in lift coefficient through changing the airfoil incidence without promoting a minimum pressure at any other than the design position with a consequent reduction in critical speed.

The distribution of the camber of an airfoil of given thickness determines its critical Mach number at the design lift coefficient. For desirable critical-speed characteristics the camber of an airfoil should be so distributed as to allow the design lift to be carried with the minimum possible reduction in critical speed below that of the corresponding symmetrical profile at zero lift. Moreover, the lift distribution should be such as to place the design lift coefficient near the middle of the lift-coefficient range over which the highest critical Mach numbers are to be obtained. The uniform load type of mean camber line was devised as a positive step in this direction for by distributing the design lift uniformly over the chord of an airfoil, local induced velocity increments were minimized and hence the reduction in airfoil critical speed below that of the basic thickness form at zero lift was small. A more promising development in this respect, however, appears to be an airfoil designed for zero load from the leading edge to the position of minimum pressure with all its lift being carried from that point to the trailing edge. An airfoil cambered in this manner would have the same critical Mach number at the design lift coefficient as the corresponding symmetrical profile at zero lift and, at the same time, the range of lift coefficient for high critical speeds would be unimpaired.

An important disadvantage associated with the rearward loading of an airfoil is the resultant large negative pitching moment. It would therefore appear to be advantageous from the standpoint of over-all airfoil characteristics to effect a compromise between the uniform type of loading and the exclusively rearward type of loading by distributing a portion of the design lift uniformly over the airfoil forward of the minimum pressure position and the remainder increasingly over the rear part of the airfoil from the minimum pressure point to the trailing edge. To investigate experimentally the effectiveness of this method for obtaining more favorable critical-speed characteristics without seriously affecting the principal aerodynamic characteristics of an airfoil, tests of an NACA 6-series airfoil section (modified NACA $66_{(109)}$-210) cambered in the manner just described were conducted in the Ames 1- by $3\frac{1}{2}$-foot high-speed wind tunnel.

The tests were confined to measurements of the section characteristics of lift, drag, and pitching moment over a velocity range from 0.3 to approximately 0.9 Mach number. The aerodynamic characteristics which profoundly influence high-speed airplane performance were evaluated and compared with the corresponding characteristics for an NACA 66-210 airfoil section having a uniform-load type of mean camber line. Mach numbers of lift and drag divergence rather than critical speeds were used as measures of aerodynamic performance at high speeds.

SYMBOLS

c_d section drag coefficient

c_l section lift coefficient

c_{l_i} design section lift coefficient

$c_{m_{c/4}}$ section moment coefficient about quarter-chord point

c airfoil chord

M Mach number

V free-stream velocity

v local velocity on the airfoil surface

x distance along chord

α angle of attack

AIRFOIL DERIVATION

The airfoil was derived from a combination of an NACA 66(109)-010 basic thickness form with a modified trailing-edge region and a camber distribution obtained as a combination of two basic NACA mean lines. The modification to the trailing edge consists of a straight-line fairing of a normal NACA 66(109)-010 airfoil to give a finite trailing-edge thickness and a continuously changing curvature from 80 percent chord forward to the 65-percent chord point where the fairing coincides with the original NACA profile. The mean camber line consists of the superposition of an a = 0.6 mean line for a design lift coefficient of -0.4 upon an a = 1.0 mean line for a design lift coefficient of 0.6. The resultant complete airfoil designation in NACA notation is as follows:

$$\text{Modified NACA } 66(109)\text{-}210 \left\{ \begin{array}{ll} a = 1.0, & c_{l_1} = 0.6 \\ a = 0.6, & c_{l_1} = -0.4 \end{array} \right\}$$

similar low-drag range

Ordinates of the modified airfoil are given in table I.

A comparison of the shape of the modified NACA 66(109)-210 airfoil together with its corresponding theoretical pressure distribution at the design lift coefficient with the shape and pressure distribution at the same lift coefficient for the NACA 66-210 airfoil having a uniform-load type (a = 1.0) of mean camber line is given in figure 1.

It should be noted that, although the respective thickness distributions near the trailing edges of the two airfoils are different, unpublished data on file at the Ames Laboratory indicate no significant differences in the characteristics at high Mach numbers of airfoils having similar differences in trailing-edge-thickness distributions. Any differences in characteristics, then, between the modified NACA 66(109)-210 and the NACA 66-210 airfoils can logically be attributed to the difference in type of camber.

APPARATUS AND TESTS

The tests were conducted in the Ames 1- by $3\frac{1}{2}$-foot high-speed wind tunnel, a low-turbulence, two-dimensional-flow wind tunnel powered by two 1000-horsepower motors. This power is sufficient to obtain the choked-flow condition discussed in reference 1 with any size model.

A 6-inch-chord model of the NACA 66(109)-210 airfoil with a modified thickness distribution and mean camber line was constructed of duralumin for the investigation. The airfoil was mounted, as illustrated in figure 2, so as to span completely the 1-foot width of the tunnel test section. End leakage was prevented, and two-dimensional flow thereby assured, through the use of sponge-rubber gaskets compressed between the model ends and the tunnel side walls.

Measurements of lift, drag, and quarter-chord pitching moment were made simultaneously at Mach numbers from 0.3 to as high as 0.9 with the airfoil at angles of attack from -6° to 16° by increments of 2°. This range of angles was sufficient to encompass the lift stall up to Mach numbers of the order of 0.8. The Reynolds numbers varied from approximately 1×10^6 at the lowest speeds to approximately 2×10^6 at the maximum speeds of the tests.

Lift and pitching moments were determined by a method similar to that described in reference 2 from measurements of the reactions on the tunnel walls of forces experienced by the airfoil. Drag was determined from wake-survey measurements made with a rake of total-head tubes.

RESULTS AND DISCUSSION

Section lift, drag, and quarter-chord pitching-moment coefficients are presented as functions of Mach number at constant angles of attack in figures 3, 4, and 5, respectively, for the modified NACA 66(109)-210 airfoil. Corresponding characteristics, obtained from earlier tests in the same wind tunnel, for the NACA 66-210 airfoil section with a uniform-load type of mean camber line are shown in figures 6, 7, and 8 for comparison. All data have been corrected for tunnel-wall interference by the methods of reference 1. The broken lines in the airfoil characteristic curves of figures 3 to 10 are used to indicate that data obtained in the vicinity of the wind-tunnel choking Mach number are not considered reliable.

Lift Characteristics

The variation in section lift coefficient with Mach number for the modified NACA 66(109)-210 airfoil is very similar to that for the NACA 66-210 profile. The corresponding cross plots (figs. 9 and 10, respectively, for the two airfoils) of the variation in section lift coefficient with angle of attack for various Mach numbers indicates the modified NACA 66(109)-210 airfoil to be appreciably different from the NACA 66-210 airfoil only in the magnitude of the maximum lift coefficient. Up to Mach numbers approaching 0.8, the maximum lift coefficients for the modified airfoil are somewhat lower than those for the NACA 66-210 airfoil. The variation in lift-curve slope with Mach number appears in figure 11 to almost exactly parallel that for the NACA 66-210 airfoil. The variations with Mach number in the respective angles of zero lift for the two airfoils may be seen in figure 12 to be virtually the same.

The only significant difference in the supercritical-speed lift characteristics of the modified NACA 66(109)-210 and the NACA 66-210 airfoils appears from figure 13 to lie in the lift-divergence Mach numbers. The Mach number of lift divergence for a given angle of attack is defined as the lowest value of the Mach number corresponding to an inflection point on the curve of lift coefficient as a function of Mach number. For all positive lift coefficients the Mach numbers of lift divergence for the modified NACA 66(109)-210 airfoil exceed those for the NACA 66-210 airfoil, the gain amounting to about 0.015 Mach number for lift coefficients ranging from the design value of 0.2 to approximately 0.85. This increment is somewhat greater than the difference (approx. 0.01) in the corresponding estimated critical speeds (taken from reference 2) for the two sections. For negative lift coefficients, however, the divergence characteristics for the two airfoils are seen to be reversed, the normally cambered NACA 66-210 airfoil having the higher divergence velocities.

Drag Characteristics

The drag characteristics of the modified NACA 66(109)-210 airfoil in general do not differ sensibly from those of comparable normally cambered airfoils. In figure 14 the variation in section drag coefficient with Mach number at zero incidence for the modified airfoil is seen to closely parallel that for the NACA 66-210 airfoil. The Mach number of drag divergence is loosely defined for present purposes as that value of Mach number at which the abrupt increase in drag coefficient commences. Beyond the drag-divergence Mach number, however, the modified NACA $66_{(109)}$-210 airfoil appears to hold a small advantage over the latter airfoil.

For lift coefficients from 0.1 to 0.6, figure 13 shows the drag-divergence Mach numbers for the modified airfoil to be higher than those for the NACA 66-210 airfoil. Throughout most of this range the difference amounts to approximately 0.01 Mach number and corresponds to the previously mentioned difference in the critical Mach numbers of the airfoils. As was noted in the case of lift divergence, the modified NACA $66_{(109)}$-210 airfoil is inferior to the NACA 66-210 airfoil in the matter of drag divergence at negative lift coefficients.

Pitching-Moment Characteristics

The variation in section quarter-chord pitching-moment coefficient with Mach number, shown in figure 5 for the modified NACA $66_{(109)}$-210 airfoil, resembles that illustrated in figure 8 for the NACA 66-210 section. Figure 15 depicts the behavior of pitching-moment coefficient with Mach number at the design lift coefficient for both airfoils. The value of the pitching-moment coefficient before divergence is, as would be expected, more negative for the rearward loaded airfoil than for a similar airfoil with a uniform-load type ($a = 1.0$) of camber line.

CONCLUSIONS

From the results of two-dimensional high-speed wind-tunnel tests of a modified NACA $66_{(109)}$-210 airfoil with a mean camber line designed to give critical speeds higher than those attainable with the uniform-load mean line, the following conclusions are drawn:

1. The Mach numbers of lift divergence for the modified NACA $66_{(109)}$-210 airfoil over most of the positive lift coefficient range are higher than the divergence Mach numbers for the NACA 66-210 airfoil with uniform-load type of camber by an amount (approximately 0.015 Mach number) roughly corresponding to the difference in the critical Mach numbers of the two airfoil sections.

2. The characteristics of lift-curve slope and zero-lift incidence for the modified NACA 66(109)-210 airfoil and for the NACA 66-210 (a = 1.0) airfoil are virtually the same.

3. The drag-divergence Mach numbers for the modified NACA 66(109)-210 airfoil are higher than those for the NACA 66-210 (a = 1.0) airfoil over a limited lift-coefficient range by an amount equal to the difference (0.01 Mach number) in the critical Mach numbers of the two profiles.

4. Pitching-moment coefficients are more negative for the modified NACA 66(109)-210 airfoil than those for the NACA 66-210 (a = 1.0) airfoil; the respective variations in this parameter with Mach number for the two sections are similar, however.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif., July 1947

REFERENCES

1. Allen, H. Julian, and Vincenti, Walter G.: Wall Interference in a Two-Dimensional-Flow Wind Tunnel with Consideration of the Effect of Compressibility. NACA ARR No. 4K03, 1944.

2. Abbott, Ira H., von Doenhoff, Albert E., and Stivers, Louis S. Jr.: Summary of Airfoil Data. NACA ACR No. L5C05, 1945.

TABLE I.– MODIFIED NACA $66_{(1.09)}$–210 $\left\{\begin{matrix} a = 1.0, & c_{l_i} = 0.6 \\ a = 0.6, & c_{l_i} = -0.4 \end{matrix}\right\}$

[Stations and ordinates in percent of airfoil chord]

Upper Surface		Lower Surface	
Station	Ordinate	Station	Ordinate
0	0	0	0
.474	.783	.526	–.743
.722	.944	.778	–.888
1.220	1.187	1.280	–1.101
2.468	1.590	2.532	–1.450
4.968	2.205	5.032	–1.973
7.469	2.687	7.531	–2.387
9.971	3.095	10.029	–2.741
14.975	3.752	15.025	–3.310
19.979	4.251	20.021	–3.751
24.983	4.634	25.017	–4.092
29.985	4.925	30.015	–4.349
34.986	5.135	35.014	–4.527
39.985	5.271	40.015	–4.633
44.980	5.336	45.020	–4.662
49.972	5.333	50.028	–4.611
54.955	5.265	55.045	–4.471
59.914	5.131	60.086	–4.209
64.886	4.891	65.114	–3.732
69.901	4.530	70.099	–3.102
74.934	4.059	75.066	–2.393
79.972	3.510	80.028	–1.690
85.009	2.901	84.991	–1.039
90.038	2.188	89.962	–.490
95.049	1.326	94.951	–.090
100.022	.077	99.978	–.077
Leading-edge radius: 0.650			

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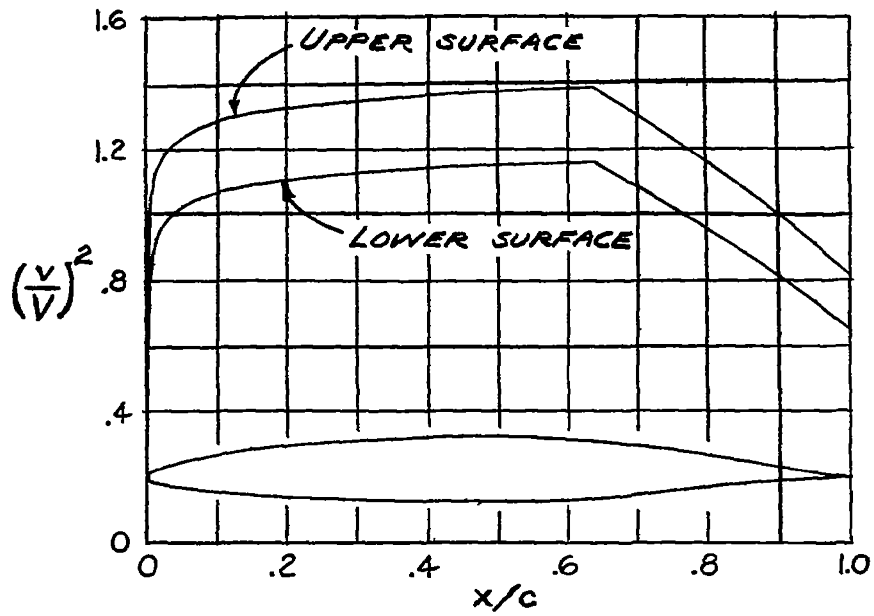

NACA 66-210 a = 1.0

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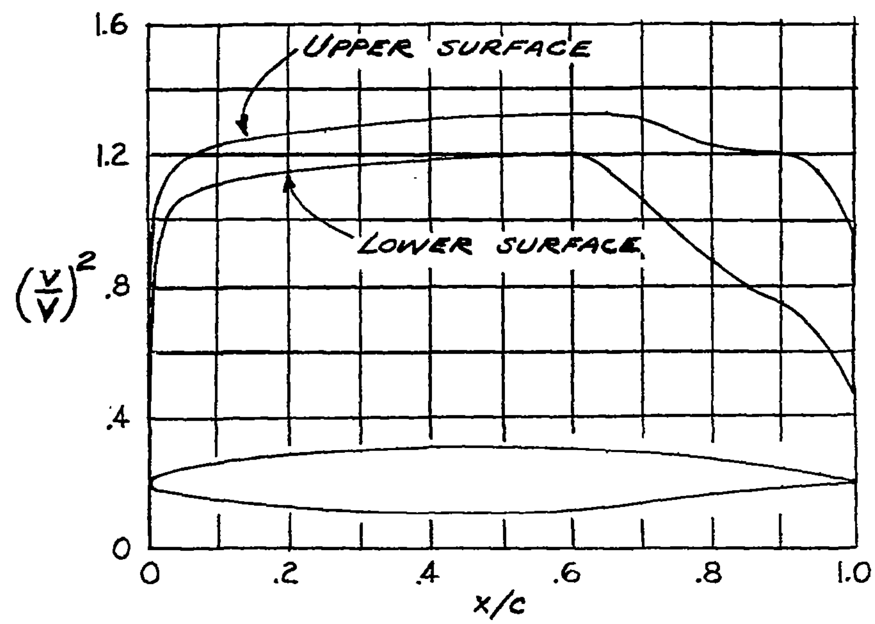

MODIFIED NACA $66_{(109)}$-210 $\left\{\begin{matrix} a = 1.0, & c_{l_i} = 0.6 \\ a = 0.6, & c_{l_i} = -0.4 \end{matrix}\right\}$

FIGURE 1.- COMPARISON OF AIRFOIL PROFILES AND PRESSURE DISTRIBUTIONS.

Figure 2.- Airfoil model mounted in the test section of the Ames 1- by 3-1/2 foot high-speed wind tunnel.

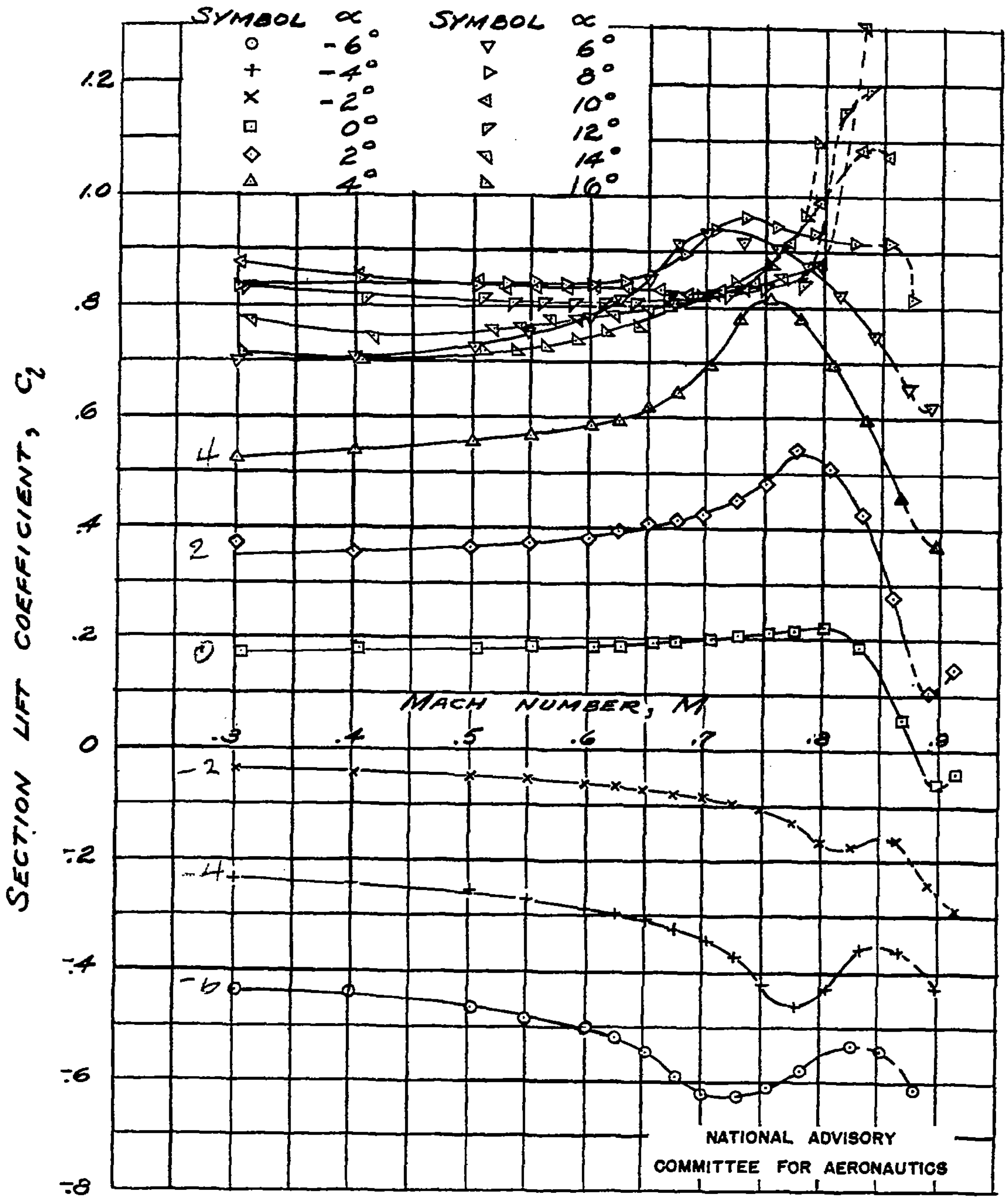

FIGURE 3 - THE VARIATION OF SECTION LIFT COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE MODIFIED NACA $66_{(109)}$-210 AIRFOIL.

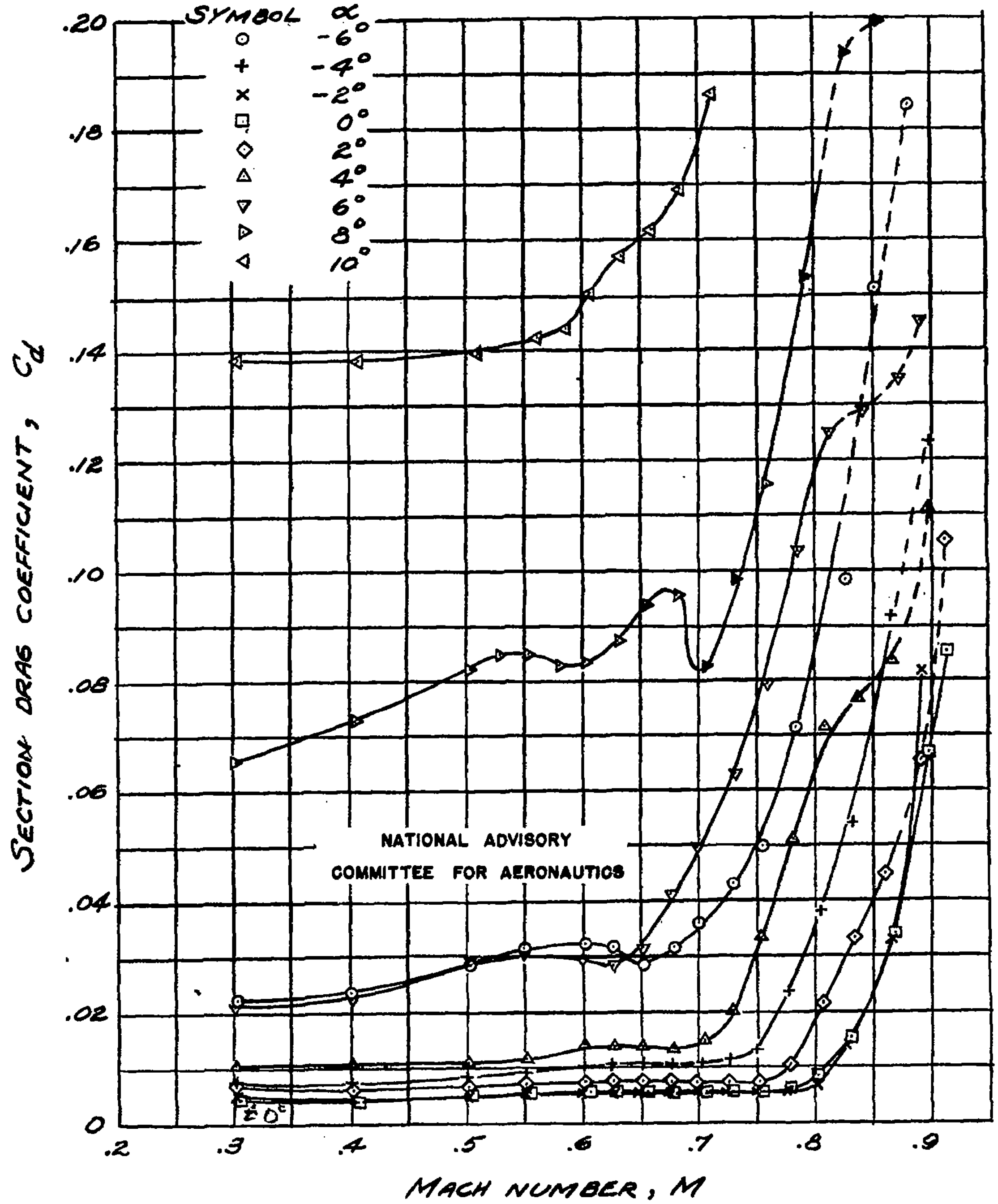

FIGURE 4.- THE VARIATION OF SECTION DRAG COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE MODIFIED NACA $66_{(109)}$-210 AIRFOIL.

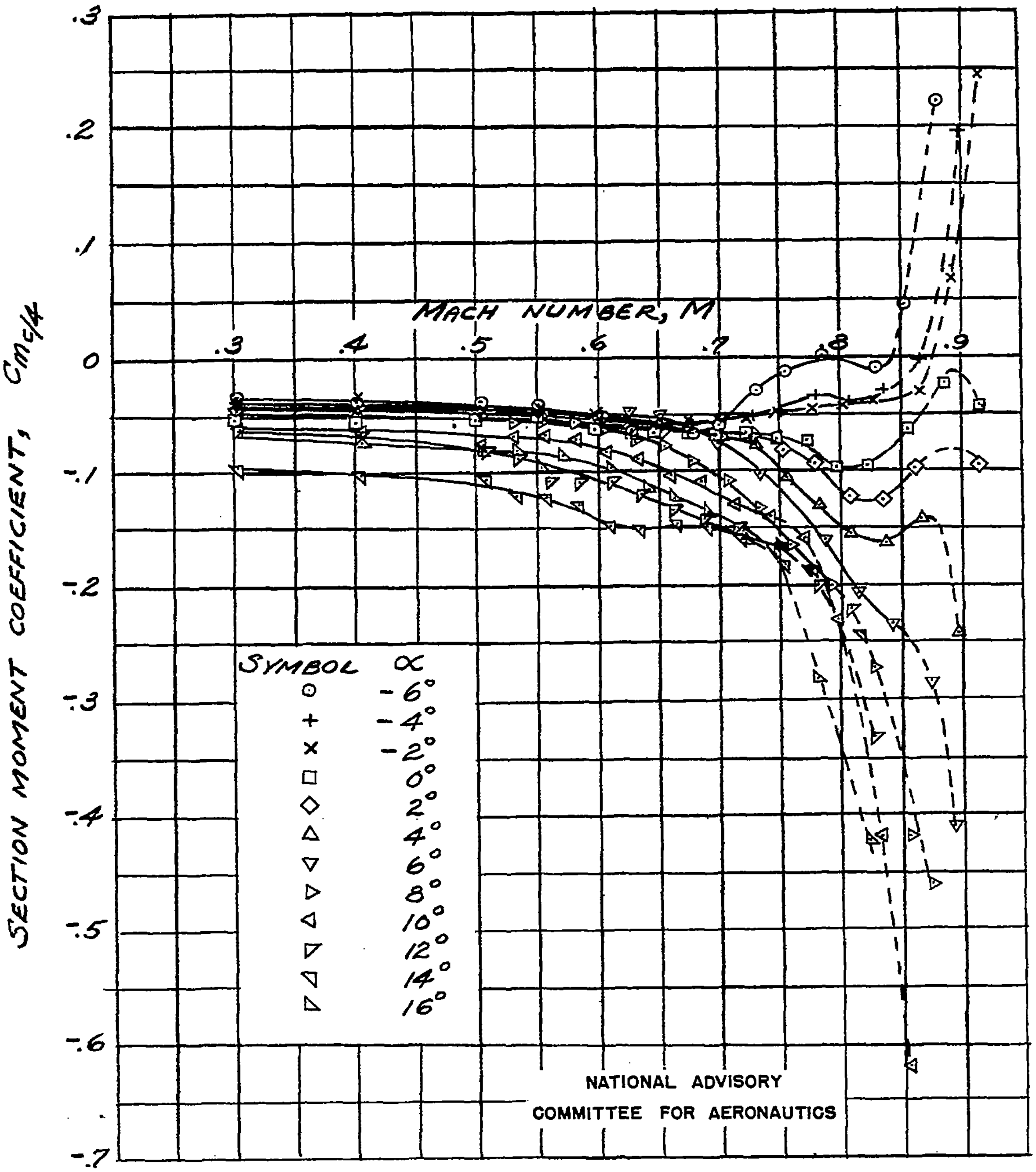

Figure 5.- The variation of section moment coefficient with Mach number at various angles of attack for the modified NACA $66_{(109)}$-210 airfoil.

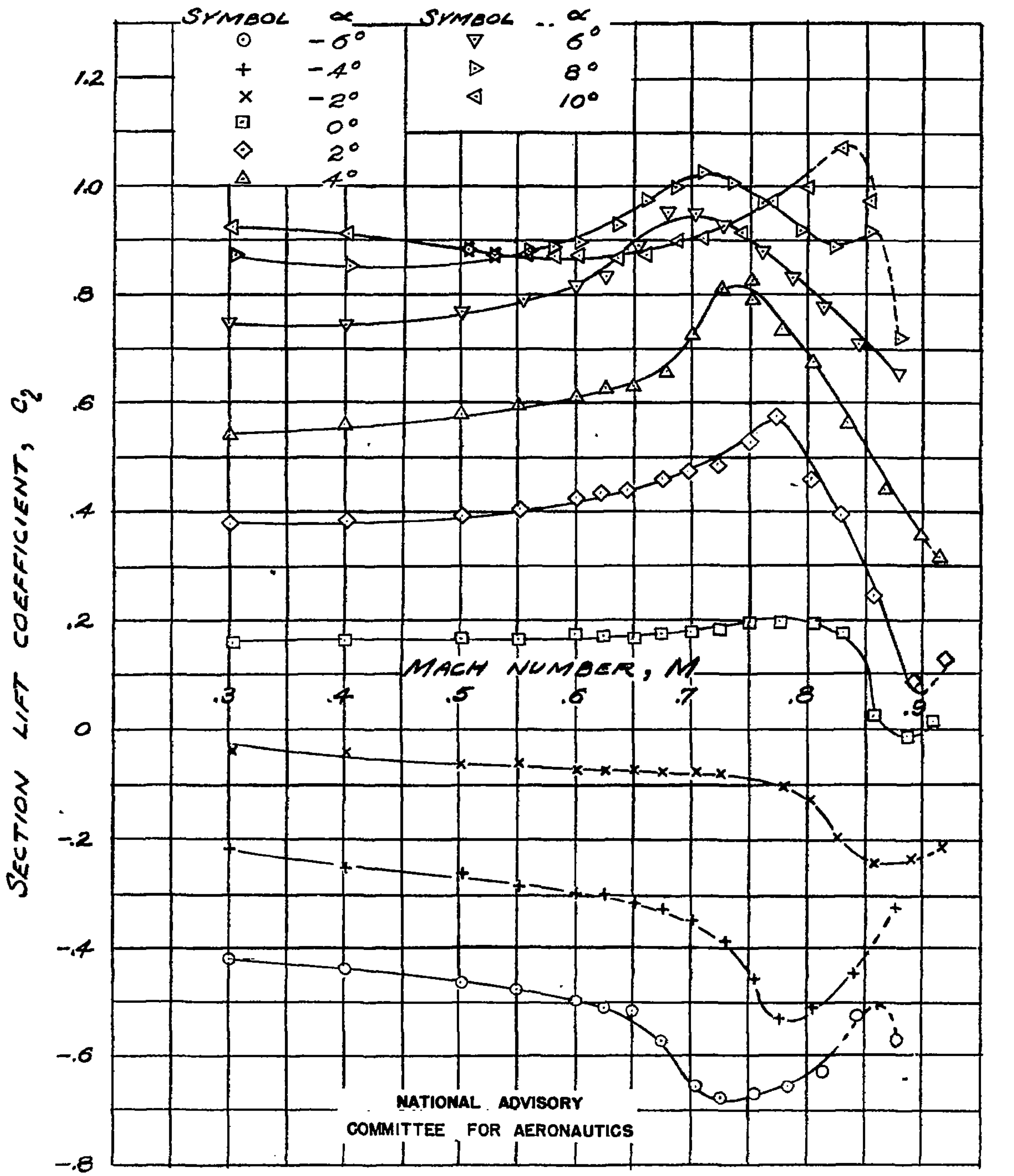

FIGURE 6.- THE VARIATION OF SECTION LIFT COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE NACA 66-210 AIRFOIL.

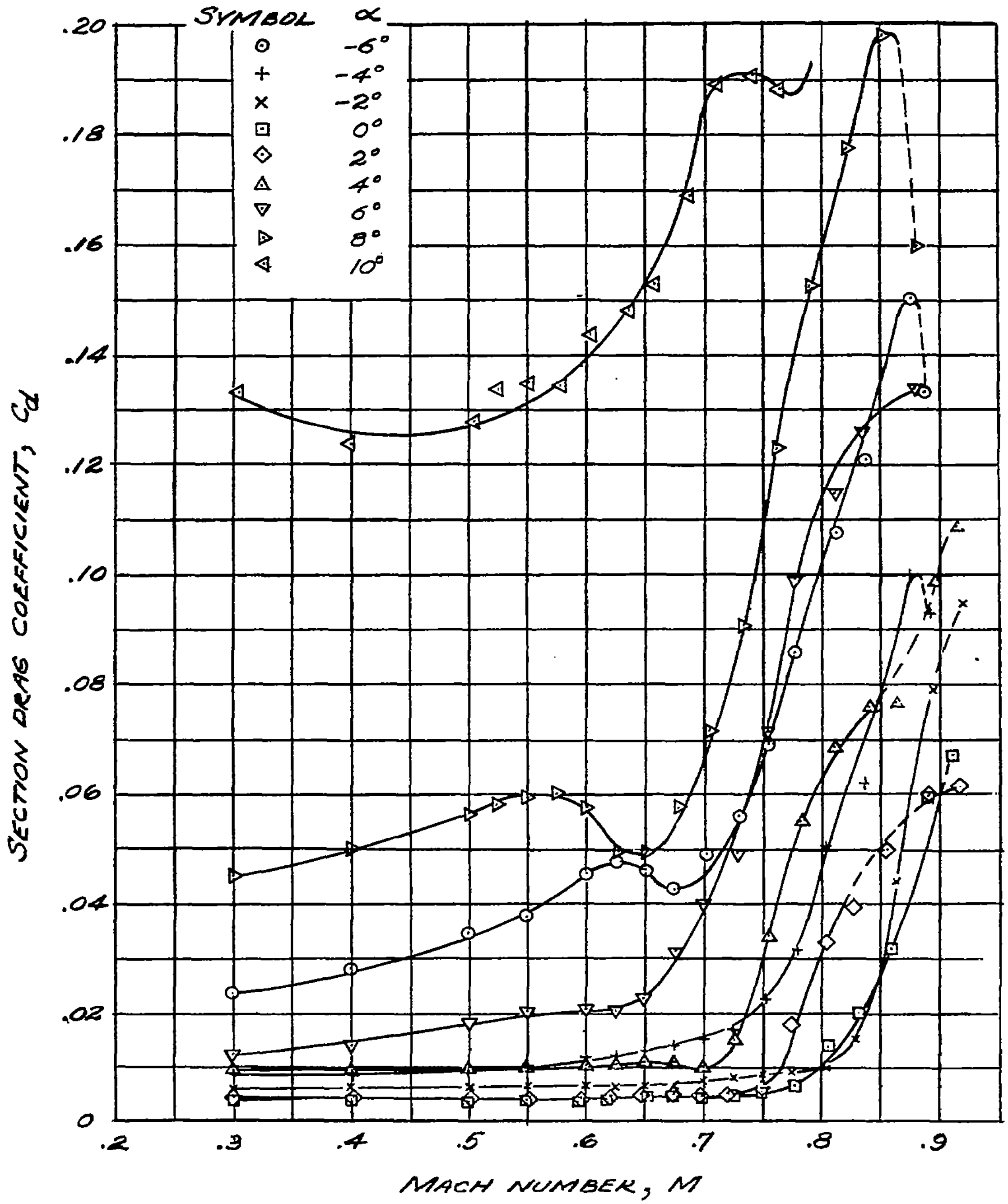

FIGURE 7.- THE VARIATION OF SECTION DRAG COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE NACA 66-210 AIRFOIL.

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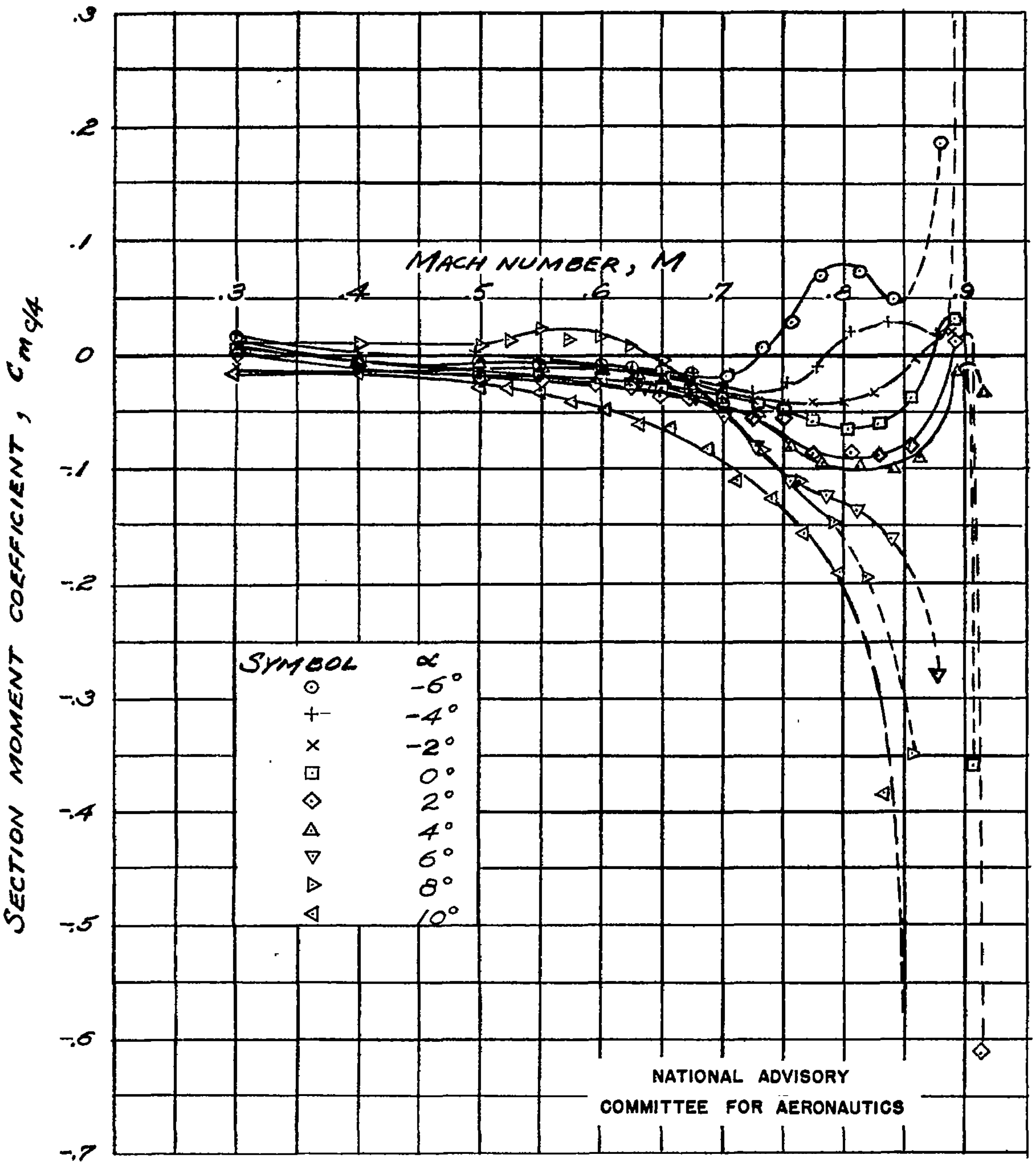

FIGURE 8.- THE VARIATION OF SECTION MOMENT COEFFICIENT WITH MACH NUMBER AT VARIOUS ANGLES OF ATTACK FOR THE NACA 66-210 AIRFOIL.

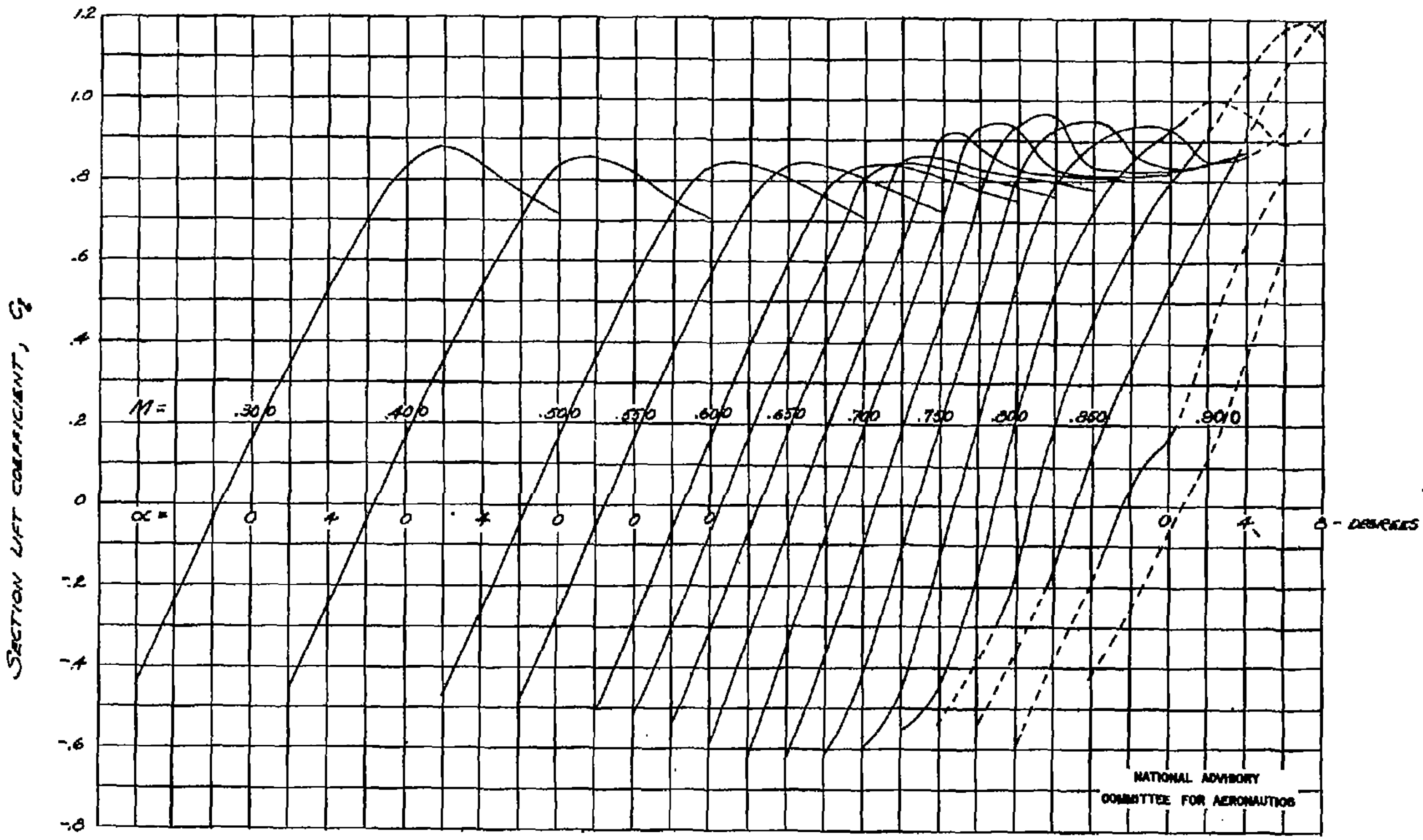

FIGURE 8.- THE VARIATION OF SECTION LIFT COEFFICIENT WITH ANGLE OF ATTACK AT VARIOUS MACH NUMBERS FOR THE MODIFIED NACA $66_{(109)}-210$ AIRFOIL.

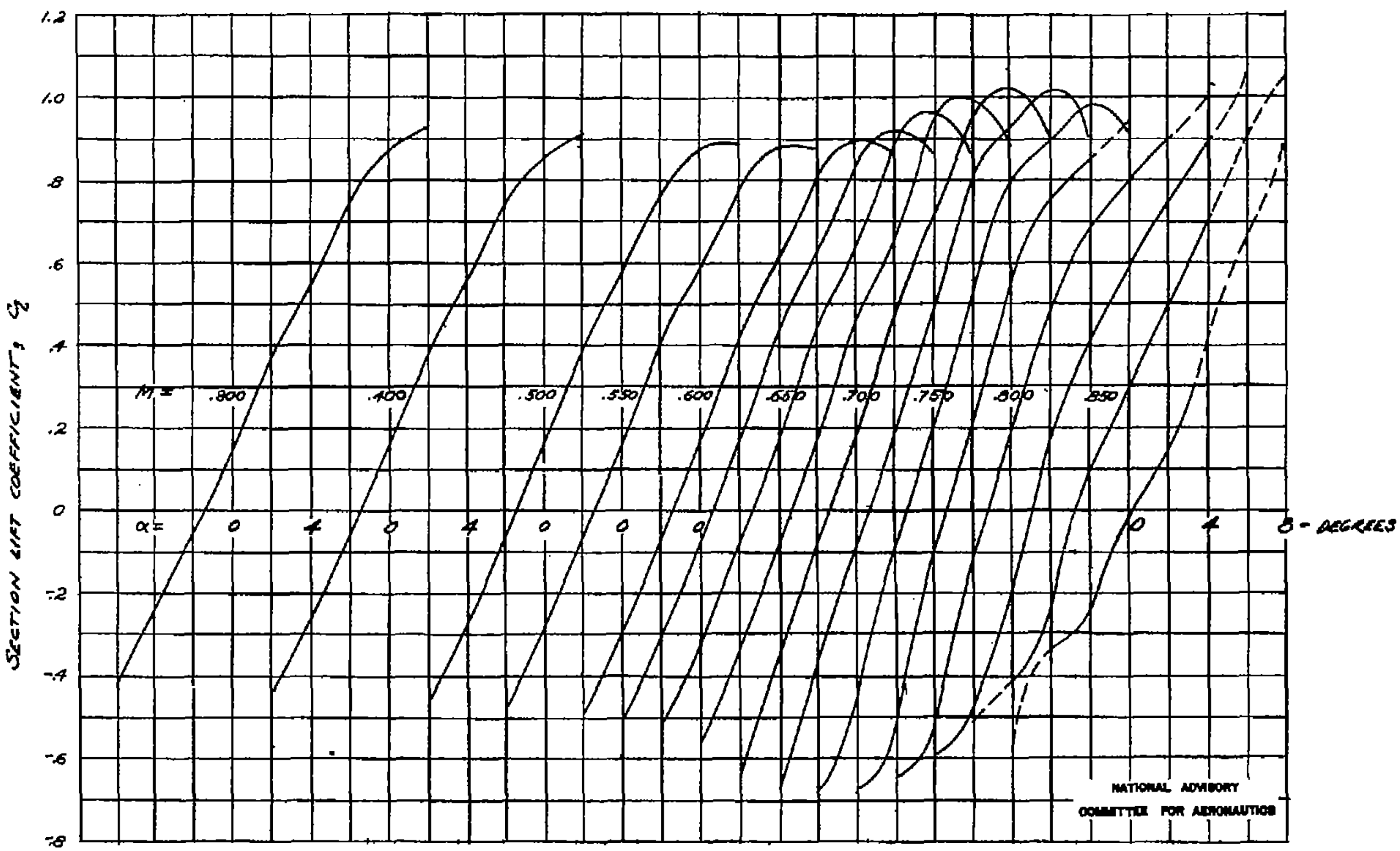

FIGURE 10:- THE VARIATION OF SECTION LIFT COEFFICIENT WITH ANGLE OF ATTACK AT VARIOUS MACH NUMBERS FOR THE NACA 66-210 a=1.0 AIRFOIL.

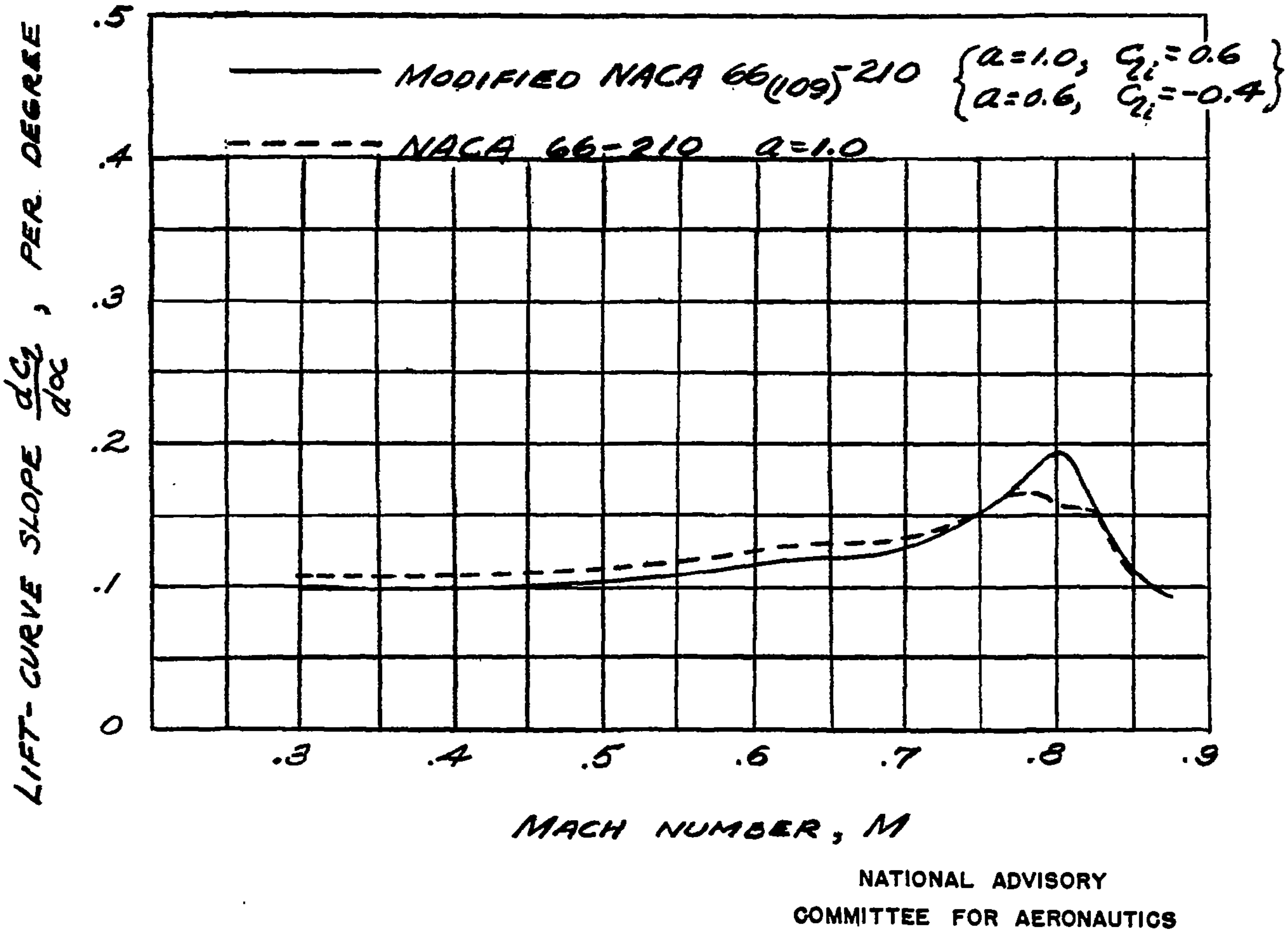

FIGURE 11.- THE VARIATION OF LIFT-CURVE SLOPE WITH MACH NUMBER AT THE DESIGN LIFT COEFFICIENT FOR THE MODIFIED NACA $66_{(109)}$-210 AND THE NACA 66-210 AIRFOILS.

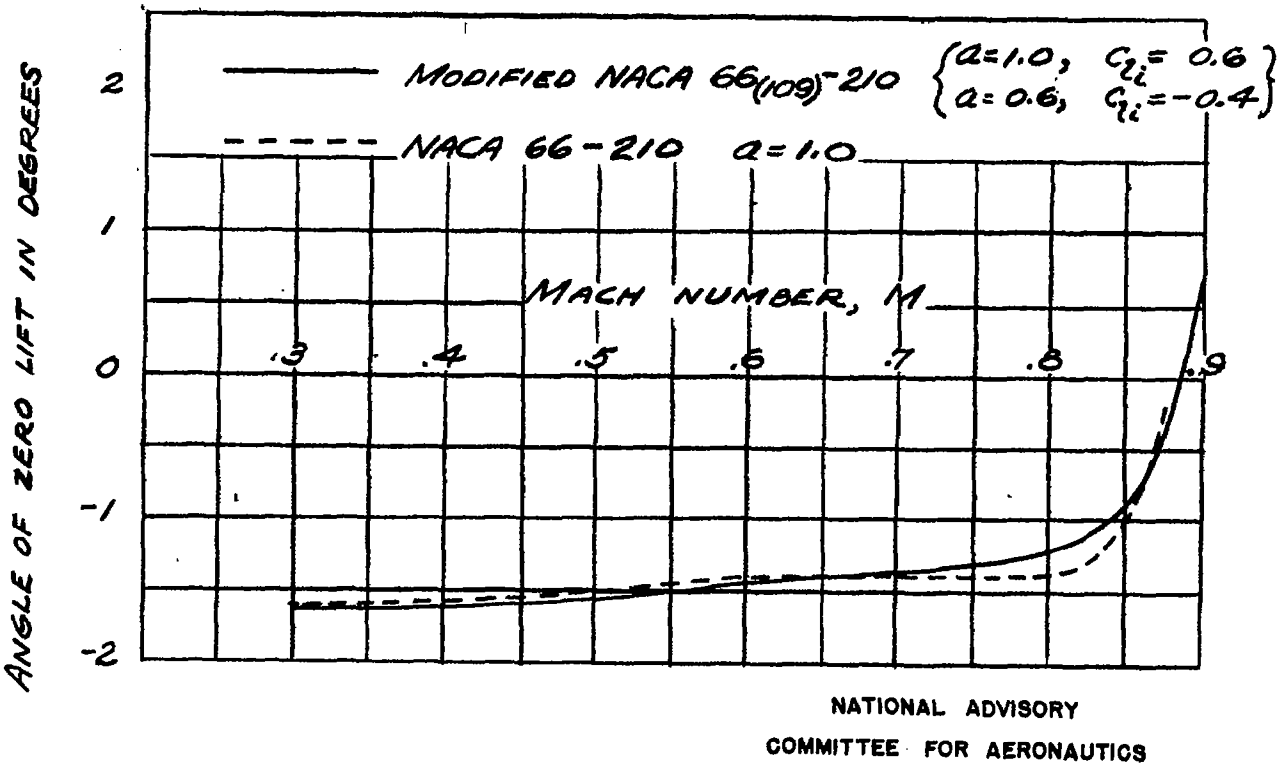

FIGURE 12.- THE VARIATION OF THE ANGLE OF ZERO LIFT WITH MACH NUMBER FOR THE MODIFIED NACA 66(109)-210 AND THE NACA 66-210 AIRFOILS.

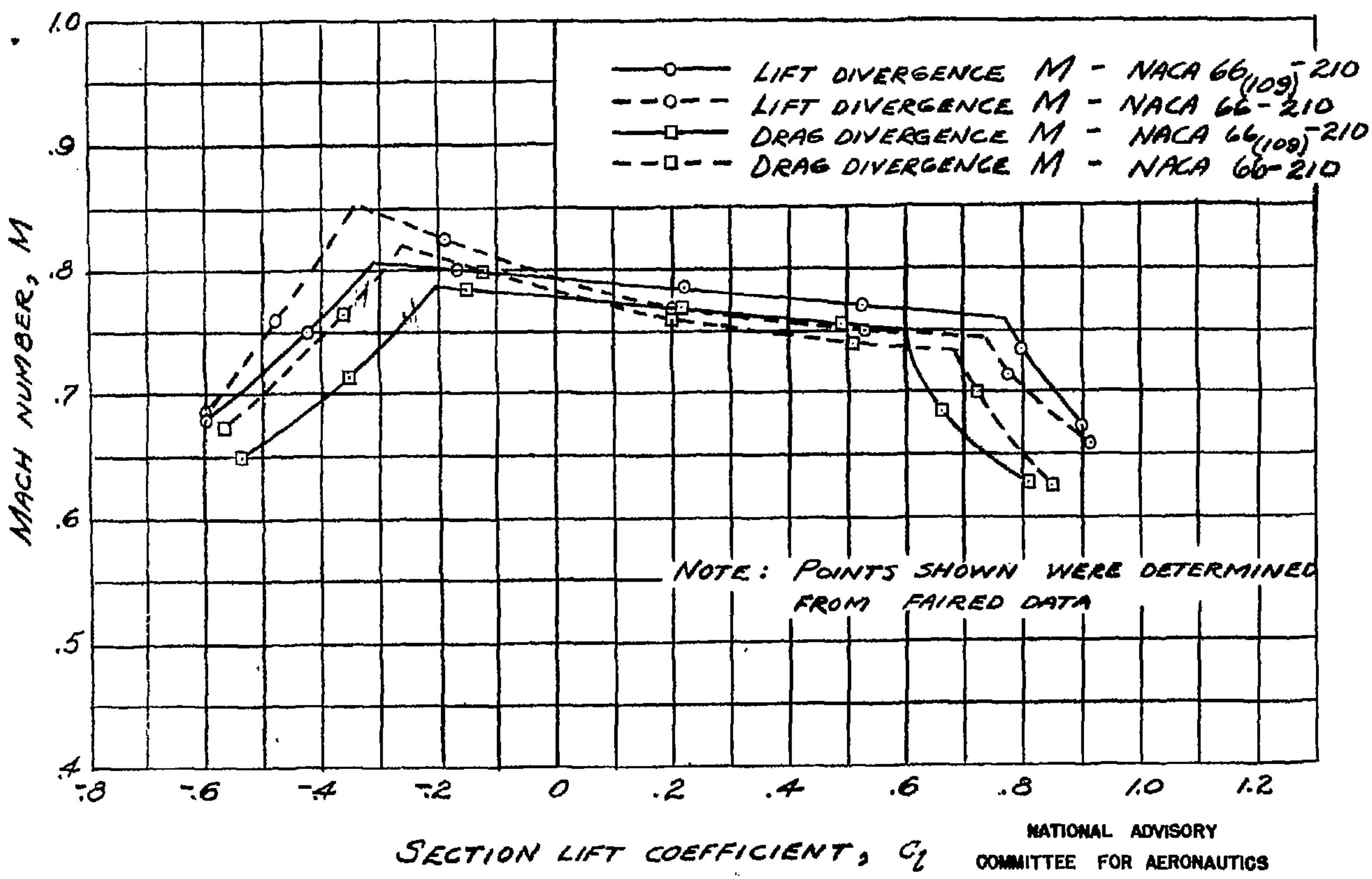

FIGURE 13 - THE VARIATION OF LIFT AND DRAG DIVERGENCE MACH NUMBERS WITH SECTION LIFT COEFFICIENT FOR THE MODIFIED NACA 66(109)-210 AND THE NACA 66-210 AIRFOILS.

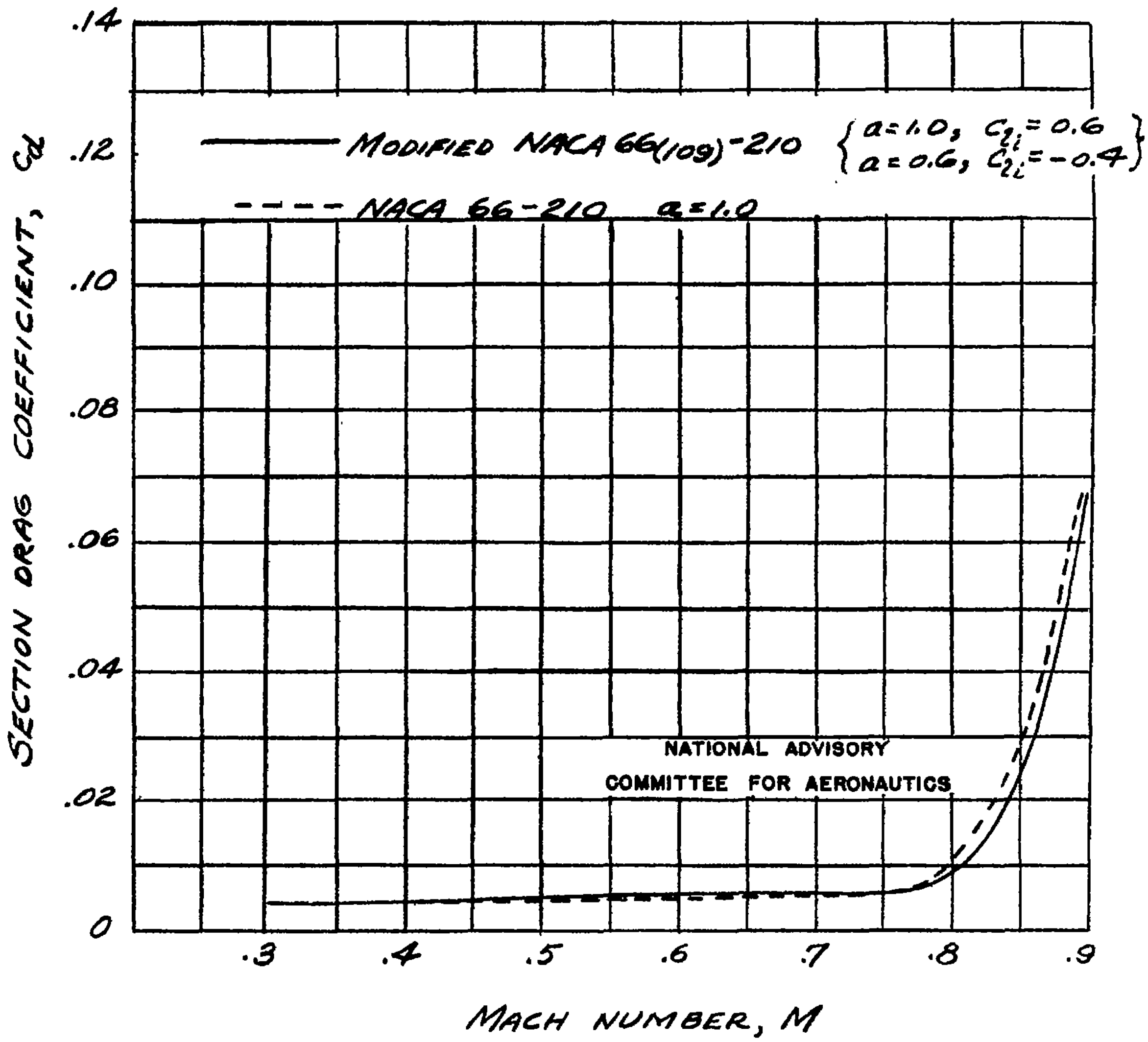

FIGURE 14.– THE VARIATION OF SECTION DRAG COEFFICIENT WITH MACH NUMBER AT ZERO INCIDENCE FOR THE MODIFIED NACA $66_{(109)}$-210 AND THE NACA 66-210 AIRFOILS.

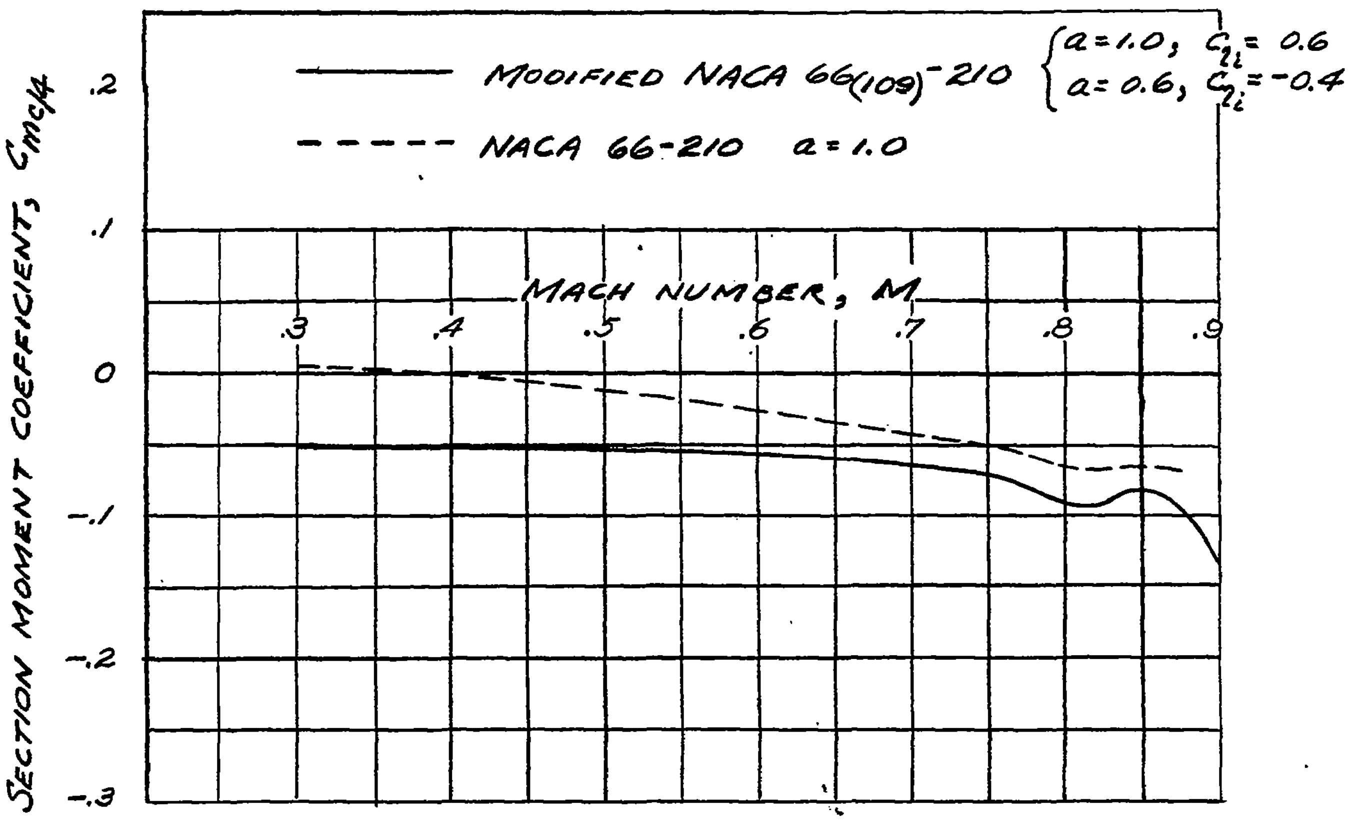

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FIGURE 15.- THE VARIATION OF MOMENT COEFFICIENT WITH MACH NUMBER AT THE DESIGN LIFT COEFFICIENT FOR THE MODIFIED NACA 66(109)-210 AND THE NACA 66-210 AIRFOILS.